

METHODS FOR KEEPING A CONDUCTIVE SPACECRAFT NEAR THE PLASMA POTENTIAL

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ABSTRACT

The floating potential of a conductive satellite can be very positive under conditions with sun-light and a very tenuous plasma such as that found in the geomagnetic tail. In eclipse the floating potential may go very negative. Different methods for spacecraft potential control are described for the purpose of carrying out low energy plasma measurements. Method 1 makes use of a hollow cathode Xe plasma emitter which makes it possible to control a spacecraft both positive and negative relative to the plasma in sunlight as well as eclipse. Method 2 uses a high energy ion emitter to limit large positive potentials in sunlight. Method 3 uses a cathode to emit electrons under eclipse conditions. For completeness the method of biasing a spacecraft relative to a large body is discussed.

1. INTRODUCTION

Spacecraft in the Earth's magnetosphere normally respond to the conditions of emitted photo-electrons due to ultraviolet light and impinging ambient plasma by charging 1-20 V positive, when illuminated by the Sun, and to potentials from a few Volts to many thousands of Volts negative when in shadow. Such potentials normally degrade the quality of measurements by particle and field experiments. In particular, for a positive spacecraft, the ambient ion population with energy less than the spacecraft potential is hidden from measurements.

Large negative potentials on spacecraft, with partly insulating surfaces, have also been associated with spacecraft anomalies and failures. This is presumably due to differential charging over the spacecraft surfaces to potentials of kilovolts, with resulting arcs or discharges damaging surface materials and causing logic upsets in the space-craft control circuitry.

Experiments on Applied Technology Satellite 6 (ATS-6) in 1974 and 1976 showed that emission of a cold plasma could discharge large negative potentials on the main frame and also exposed insulating surfaces. The hollow cathode plasma source, also known as a plasma bridge neutralizer when used with ion engines, worked well in sunlight and eclipse. It was

also capable of reducing positive potentials from +10V to $\pm 2V$. The resulting spacecraft potential was found to depend upon the potential of the plasma source (particularly the anode, or keeper). Control of the anode potential, therefore, provides the ability to control the spacecraft potential (Ref. 1).

The ATS-6 experiments demonstrated the feasibility of using a cold plasma source for control of the conductive main frame of the satellite. In this paper we will consider the use of a cold plasma source and other methods to control the potential of a fully conductive spacecraft. A conductive surface is obviously a prerequisite for having a well defined potential and potential control.

The GEOS and ISEE spacecraft all had conductive surfaces and it was possible to determine their potentials in sunlight by using as a reference an electric field double probe, with sensors biased to be near the plasma potential (Refs. 2 and 3). The spacecraft potential could also under certain conditions be determined, in sunlight as well as eclipse, by measurements of shifts of particle energy spectra. Neither of these spacecraft had any active control for bringing their potential closer to that of the plasma. The floating potentials observed in different plasma regimes nevertheless is an important starting point for discussing potential control.

2. CHARACTERISTIC FLOATING POTENTIALS AND INFLUENCE ON PLASMA MEASUREMENTS

In the following table the floating potential V_f is given for a conductive satellite in different plasma environments. The numbers are based on GEOS and ISEE observations.

Plasma Environment	$V_f(\text{sunlight})$	$V_f(\text{eclipse})$
Solar Wind	+ (5 to 10) V	NA
Magnetosheath	+ (2 to 5) V	NA
Outer Magnetosphere	+ (2 to 15) V	- (5 to 1000) V
Lobes	+ (15 to 100) V	NA
Plasmasphere	+ (0.1 to 2.0) V	- (0.1 to 1.0) V

The negative charging to -1000 V is the most extreme negative voltage observed by accelerated ions on GEOS-2 in geostationary orbit and in eclipse (Private communication, G. Wrenn and D. Young). ISEE-3 was after the launch in 1978 placed in an orbit near the Sunward Lagrangian point, and was in 1982 moved to a number of magnetotail orbits before injection, in December 1983, in a new orbit taking the spacecraft to comet Giacobini Zinner. Charging to positive potentials approaching 100 V has been observed on ISEE-3 by measurements of accelerated electrons in the lobes of the magneto-tail at distances of the order 3 times the distance from the Earth to the Moon (Private communication, J. Gossling).

The positive floating potential in sunlight is determined by the emitted photoelectron distribution and the distribution of collected ambient electrons in such a way that the integrated flux of emitted and collected electrons balance each other. The contribution from ions is negligible except for the solar wind. It then follows that the floating potential is more positive if the plasma becomes more tenuous. A very positive spacecraft makes low energy electron measurements difficult because such measurements are masked by photo-electrons returning to the spacecraft, up to the energy corresponding to the spacecraft potential. Ion measurements are not possible for energies below that corresponding to the spacecraft potential.

In eclipse electron measurements below the energy corresponding to the spacecraft potential are impossible because these electrons will be repelled by the negative spacecraft.

In practice the most common problem is large positive spacecraft potentials in sunlight because spacecraft spend most of, or all their time in sunlight.

3. SPACECRAFT POTENTIAL CONTROL SYSTEMS

3.1. The Hollow Cathode Cold Plasma Emitter

The concept of this plasma source is based on the establishment of a dilute plasma cloud that is composed of relatively low-energy electrons and ions. The ions in the plasma enable the transport of electrons through space, to neutralize charge imbalance, without requiring the establishment of high potential gradients. This is the concept on which the plasma-bridge neutralizer (used on ion thrusters) is based. These neutralizers have been used in space, for example in connection with ATS-6 ion thruster experiments. In these applications, a hollow cathode discharge has been used to generate the plasma required for electron transport.

Figure 1 shows how the operation of such a plasma source in sunlight makes it possible to bring a spacecraft to any potential, near the plasma potential. The upper part of the figure shows the source kept at the same potential as the spacecraft. The separate body for the source is meant to illustrate that the plasma source must protrude at some distance from the spacecraft in order to be efficient. The

exact geometry requires studies in laboratory plasmas. In the case that the plasma source is not operated the spacecraft will float at a potential V_f determined by the collected ambient electron current I_{ae} and the emitted photocurrent I_p . By operating the source, the spacecraft potential V_s moves closer to the plasma potential V_{pl} due to emission of cold ions. The value of V_s is determined by; $|I_p| = |I_{ae}| + |I_{s+}|$

where I_{s+} is the ion emission current of the plasma source.

The lower part of Figure 1 shows that the spacecraft can be biased to small potential differences relative to V_{pl} . The curves, showing the current balance for the whole system are drawn on the assumption that I_p and I_{ae} are negligible for the source. Furthermore the ion current from the source to the more negative spacecraft has in this case been considered negligible. This obviously requires further study.

The operation of the same plasma source in eclipse has a similar effect. By emission of electrons V_s moves closer to V_{pl} as shown in Figure 2. The floating potential V_f , without plasma source operation, is determined by collected ambient electrons and ions, I_{ae} and I_{a+} . By turning on the plasma source an electron emission current I_{se} goes from the source. Observe that the emission of electrons from the source is more efficient and V_s is closer to V_{pl} than for the case shown in Figure 1. Again the spacecraft can be biased relative to the plasma source to be near V_{pl} . Electrons from the source tend to go to the spacecraft.

The elements required in a hollow-cathode discharge are shown schematically in Figure 3. Thermionic emission must be provided to ignite and maintain a "keep-alive" or keeper discharge between the internal surfaces of the hollow cathode and the keeper electrode. To minimize the heater power required, emission can be maintained at relatively low temperatures by employing barium to lower the cathode-surface work function. A bariated surface is provided by using a porous-tungsten-matrix reservoir or "insert" that is impregnated with barium aluminate. The insert forms the interior emitting surface in the hollow cathode discharge. By establishing appropriate operating conditions (heater power, discharge current, and expellant flow) a balance can be achieved between removal and replenishment of the surface barium thereby providing stable, long-lifetime cathode operation. In order to generate a plasma Xe gas is fed through the hollow cathode where a Xe plasma is generated and expands from the ringshaped anode. Approximately 2 kg of Xe is sufficient for several years of operation.

The plasma source will be more than capable of balancing the net negative currents found in eclipse charging events, producing 10-100 microamps of electron current (Ref. 1). The largest current that will be required is the ion current necessary to balance the total photoelectric current. For a photocurrent density of 10 to 50 $\mu A m^{-2}$ (Ref. 4), a spacecraft of 10 m^2 area requires a plasma source capable of 100 to 500 μA of ion current. The results of the ATS-6 ion engine operation (Ref. 1) and the laboratory results of Komatsu and Sellen (Ref. 5) show that a plasma source is also capable of the necessary ion current.

Figure 4 shows data from ATS-6 which demonstrates that such a source can function as described. The following numbers refer to Figure 4;

1. The source is turned on and the spacecraft is biased a few Volts negative relative to the source. The returning photo-electrons seen as a white band near 5-10 eV disappear as the satellite goes near V_{pl} or negative, i.e. photo-electrons escape instead of returning.
2. The source plasma generation is increased and the spacecraft goes more negative as seen by collection of ambient ions near 10 eV (the blackening inside the white means increased intensity). The ions could be identified as ambient ions, and not plasma source ions, because of their pitch angle distribution.
3. Start of the eclipse. The spacecraft goes slightly more negative due to disappearance of photo-emission.
4. Plasma source was turned off in eclipse and the spacecraft floats near zero and collects cold ions.
5. Start of penumbra.
6. Return to full sunlight.

3.2. High energy ion emitter

Figure 5 illustrates how the emission of high energy ions (e.g. 0.5 to 1.0 keV) can inhibit a space-craft from charging to very positive potentials in sunlight under tenuous plasma conditions. The upper part shows that V_s (satellite potential for ion emitter operation) is slightly less than V_f for a relatively dense plasma. The lower part shows that V_s goes nearly to the same value when the plasma is very tenuous and V_f is large.

For a spacecraft with 10 m^2 sunlit surface area and a photo-emission of $10\text{-}50 \text{ PA m}^{-2}$ (Ref. 4) it is sufficient to emit of the order 50 pA to assure that V_s is of the order +3 V or less.

From Figure 5 it appears to be possible to bring V_s close to V_{pl} by increasing the emitted ion current I_{s+} to be slightly smaller than the maximum photo-emission. In practice this is not feasible, because if I_{s+} by accident exceeds the maximum photocurrent, the spacecraft will swing to very negative potentials. The only reasonable approach is therefore to use a value for I_{s+} which is safe and furthermore apply current limiters and voltage limiters for negative spacecraft potentials by making reference to an electric field probe kept near the plasma potential.

It would be possible to use a conventional ion emitter with a gas supply and acceleration of ions from a discharge similar to that described in the previous section. However for the current levels required it is also possible to use a more simple and compact emitter. Furthermore it is worth mentioning that the emitter can be mounted inside a small aperture in the satellite skin, and does not need to protrude from the spacecraft.

Figure 6 shows an ion emitter which is based on an existing low level ion thruster design (Ref. 6). A suitable material is brought from solid to liquid form by heating, and the liquid is brought by capillary forces to a sharp tip placed near to a

negative, accelerating ring. Ions are formed near the tip by field emission and are accelerated away from the source.

Most of the experience is with Cesium, however other materials such as Tin and organic materials have been used. It is clearly preferable to avoid Cesium, an aggressive gas which can cause damage in case of incorrect operation of the source.

Of the order 5-20 grams of material is sufficient for several years of operation.

The ion emitter can be turned on and off by controlling the accelerating voltage and over longer periods by controlling the heater power.

Comparing the high energy emitter with the cold plasma source, the former can inhibit large positive potentials in sunlight, whereas the latter can bring the spacecraft to a predetermined potential near the plasma potential for sunlight as well as eclipse conditions.

3.4. Cold electron emitter

Previous attempts on ATS-5 to discharge a space-craft at a negative potential in eclipse were only partly successful. The negative potential was reduced, but not to any significant degree (Refs. 7 and 8). The reason for this was probably due to differential charging effects on these spacecraft and also due to the emitter being close to the surface.

In order to be effective an electron emitter, in the form of a heated filament or indirectly heated cathode, must be placed on a short boom so that there is a rapid potential drop, i.e. large electric field, near the emitter and a sufficient number of electrons can be pulled out from the source and thereby discharge the spacecraft.

Instead of using a heated electron emitter Ref. 9 has described the use of a sharp tip on a short boom in order to emit electrons by field emission. Such a system would not require power and would be selfregulating. However the efficiency of this system must be demonstrated, and initially the safest approach is probably to use a small heated emitter.

3.5. Biasing of a spacecraft relative to another conductive body

Figure 7 shows how a positively charged satellite in sunlight can be controlled to less positive potentials, even to be at the plasma potential, by a suitable bias relative to another conductor which preferably is much larger than the spacecraft and is at a distance of several Debye lengths. The connecting wire must be in the form of a coaxial cable near the spacecraft with the outer braid connected to spacecraft ground. This will avoid a positive wire near the spacecraft.

This system is described more to complete the list of possible ways of potential control. In practice it is difficult to make use of such a system because it requires complicated and costly deployment, and it will interfere with the attitude and orbit control of the spacecraft. It may be feasible to have a

partial solution in the form of a lightweight sphere, of similar size as that of the spacecraft, fixed at the end of a long boom. The spherical shape is necessary in order that the reference body does not change its photoemission with a change in orientation. The advantage with such a system is that there will be no interference from emitted ions in the vicinity of the spacecraft.

It seems difficult to use such a system in eclipse, except for cases when the reference body charges to moderate negative potentials.

4. SUMMARY

This paper presents possible methods for control of the potential of a spacecraft in order to carry out measurements on the full energy distribution of ambient electrons and ions. The negative charging of a spacecraft in eclipse has received considerably publicity, however the charging of a spacecraft to large positive potentials in a tenuous plasma in sunlight has received less attention.

The starting point for descriptions of different systems for potential control is a spacecraft with fully conductive surfaces. Any insulator, not in sunlight, can charge to negative potentials which are difficult to predict and the concept of controlling the potential of the spacecraft is not valid.

A cold Xe plasma source as described is the most powerful system in terms of control of the spacecraft to any potential near the plasma potential. This source is effective both for sunlight and eclipse conditions.

A high energy ion emitter can be used to limit positive spacecraft potentials in sunlight to be not larger than $+2$ to 3 V.

A small electron emitter (heated filament or cathode) placed on a short boom is a simple and efficient way to limit negative charging in eclipse to a few Volts negative.

A large conductive, spherical reference body at a large distance from the spacecraft can be used to bias the spacecraft to potentials near the plasma potential. The system has obvious mechanical complications which may not make it feasible.

It should be made clear that the systems described require further development and study before they can be flown on any future spacecraft. Such a development programme must address a number of problem areas such as;

1. Better understanding of the different control devices by carrying out laboratory plasma experiments.
2. Possible interference to plasma and wave experiments.
3. Make sure that no irregular operation of the devices can damage spacecraft systems or scientific instruments.

The GEOS and ISEE satellites have provided a better understanding of spacecraft plasma interactions and have provided new information on low energy electrons and ions.

However it has also become apparent that spacecraft potential control is necessary in order to carry out complete plasma measurements. This has been recognized by ongoing science working teams involved in the studies of future projects such as OPEN (NASA) and CLUSTER (ESA).

5. REFERENCES

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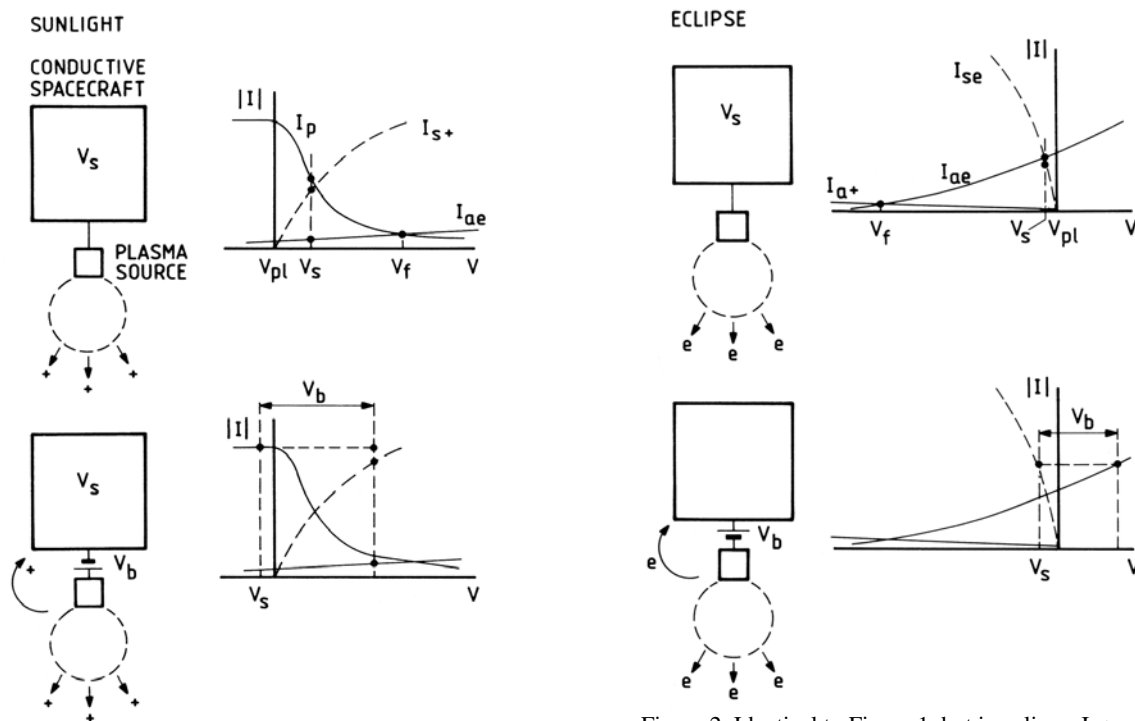


Figure 2. Identical to Figure 1, but in eclipse. I_{a+} = collected ambient ions, I_{se} = escaping source electrons.

Figure 1. Current-voltage relationship for a spacecraft with a cold plasma source in sun-light. I_p = escaping photoelectrons, I_{ae} = collected ambient electrons, I_{s+} = escaping source ions, V_{pl} = plasma potential. The spacecraft will be at a potential V_f without plasma source and will go to a potential V_s with plasma source in operation. The lower part shows that V_s can be brought to any value near V_{pl} , in this case to a negative potential, by biasing the spacecraft relative to the source.

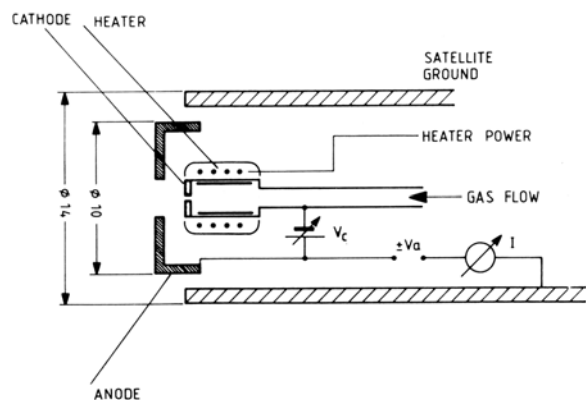


Figure 3. Hollow cathode cold plasma source.

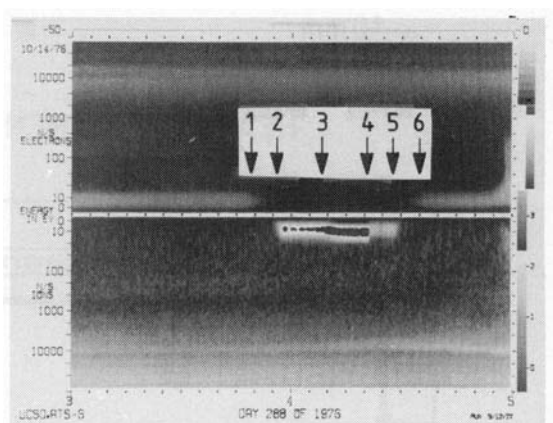


Figure 4. Electron and ion data from ATS-6 during operation of a Cesium plasma source. The events labeled with numbers are explained in the text.

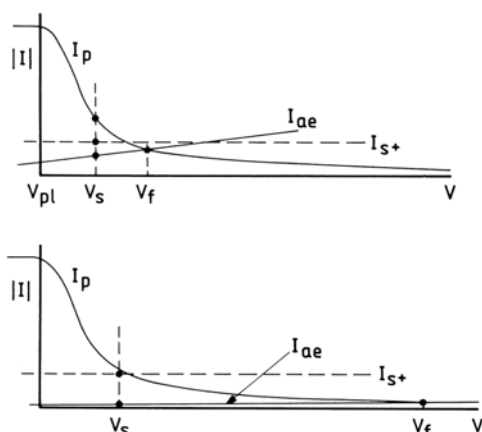


Figure 5. Current-voltage relationship for a space-craft with a high energy (keV) ion emitter. The upper (lower) part shows a relatively high (low) magnetospheric density. The ion emitter current IS+ keeps Vs at small values in both cases.

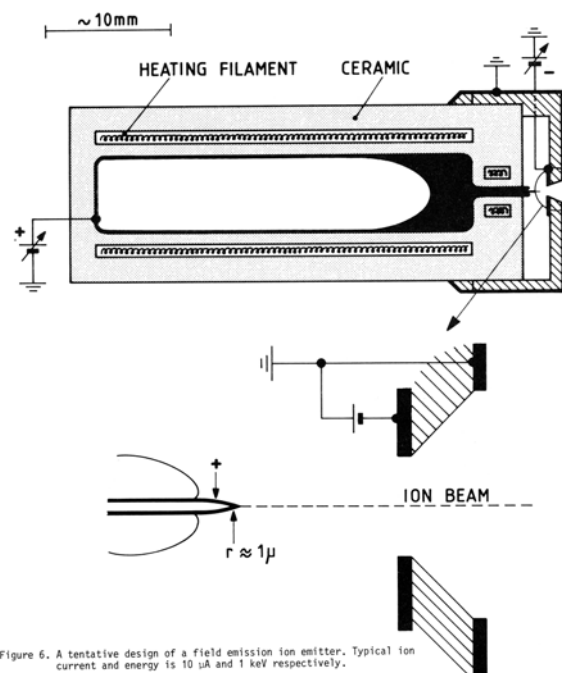


Figure 6. A tentative design of a field emission ion emitter. Typical ion current and energy is 10 μ A and 1 keV respectively.

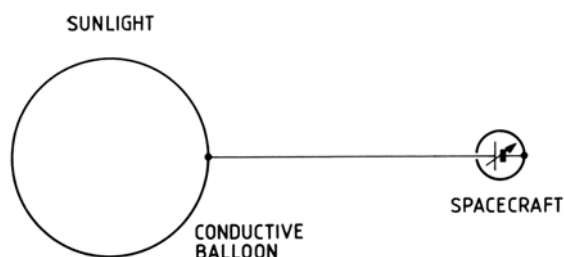


Figure 7. Biasing of a conductive spacecraft relative to a larger conductive sphere.